

NEAR EARTH ASTEROID RENDEZVOUS (NEAR) REVISED EROS ORBIT PHASE TRAJECTORY DESIGN

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Abstract

Trajectory design of the orbit phase of the NEAR mission involves a new process that departs significantly from those procedures used in previous missions. In most cases, a precise spacecraft ephemeris is designed well in advance of arrival at the target body. For NEAR, the uncertainty in the dynamic environment around Eros does not allow the luxury of a precise spacecraft trajectory to be defined in advance. The principal cause of this uncertainty is the limited knowledge of the gravity field and rotational state of Eros. As a result, the concept for the NEAR trajectory design is to define a number of rules for satisfying spacecraft, mission, and science constraints, and then apply these rules to various assumptions for the model of Eros. Nominal, high, and low Eros mass models are used for testing the trajectory design strategy and to bracket the ranges of parameter variations that are expected upon arrival at the asteroid. The final design is completed after arrival at Eros and determination of the actual gravity field and rotational state.

The application of the NEAR orbit phase trajectory design to the current best estimate of the Eros physical parameters is described in this paper. The resulting orbit is the prototype for the actual trajectory design to be performed upon arrival at Eros in February of 2000. The trajectory is

described and illustrated, and some of the problems encountered in the design and their resolution are discussed.

Introduction

As a result of the unplanned termination of the deep space rendezvous maneuver on December 20, 1998, the NEAR spacecraft passed within 3830 km of Eros on December 23, 1998. This flyby provided a brief glimpse of Eros, and allowed for a more accurate model of the rotational parameters and gravity field uncertainty [1]. Furthermore, after the termination of the deep space rendezvous burn, contact with the spacecraft was lost and the NEAR spacecraft lost attitude control. During the subsequent gyrations of the spacecraft, hydrazine thruster firings were used to regain attitude control. This unplanned thruster activity used much of the fuel margin allocated for the orbit phase. Consequently, minimizing fuel consumption is now even more important.

The deep space burn was finally executed on January 3, 1999, but this maneuver put NEAR on a trajectory that does not arrive at Eros until February 2000. Thus, the arrival conditions and orbital geometry are quite different from the previous orbit phase trajectory design [2]. As a result of these unforeseen events, the orbit phase trajectory design must now be revised to minimize fuel consumption and also to accommodate the different geometric conditions that the NEAR spacecraft will encounter at Eros in 2000. This paper discusses the revised orbit phase design process and results.

Spacecraft and Mission Constraints

Before defining a targeting strategy, it is necessary to define spacecraft and mission constraints that the spacecraft trajectory must satisfy. These constraints are then transformed to trajectory design parameters and quantified. The final step in the design process is to target a trajectory that satisfies the numerical values assigned to these target parameters.

The spacecraft constraints that apply to the Eros orbit phase design include limiting fuel consumption and maintaining solar panel illumination. Other constraints relate to the flexibility and speed that mission operations can be performed and achieving the science requirements.

Probably the most important spacecraft constraint is to fly the prime mission within the remaining propellant budget. This is an even greater priority due to the unplanned events that occurred in December.

The most challenging constraint to satisfy concerns solar panel illumination. To satisfy spacecraft power requirements, the solar panels

cannot be turned more than about 30° off the sun line [3]. Since the science instruments are fixed with respect to the spacecraft body, it is necessary to turn the spacecraft to point these instruments at Eros. If the angle between the line to nadir and the plane perpendicular to the sun line is greater than 30° , the nadir point cannot be imaged without violating the solar panel illumination constraint. A coordinate frame is defined with the z-axis pointing away from the Sun, and the equatorial or x-y plane perpendicular to the Sun-line. This is referred to as the Sun Plane-Of-Sky (POS) coordinate frame. In the Sun POS coordinate frame, orbits with inclination less than 30° direct, or greater than 150° retrograde will not violate the solar panel constraint and allow imaging of nadir from any point in the orbit.

Another important constraint relates to the time to conduct mission operations. In order to conduct the mission smoothly without resorting to around-the-clock operations, the minimum time between spacecraft propulsive maneuvers is limited to one week. The major problem is turning around accurate orbit determination solutions in time to perform the propulsive maneuvers that are required to keep the spacecraft on course. The velocity change resulting from maneuver execution errors corrupts the orbit solution and compromises the quality of science. Furthermore, additional risk to the mission is incurred because of poor trajectory control. A rapid redetermination of the orbit places a large amount of pressure on the Mission Operations team to deliver accurate solutions for the spacecraft orbit. As a result, a goal of allowing a minimum of one week between maneuvers was introduced in an attempt to minimize this pressure. This provides an increase in both data quantity and quality for the orbit solution.

Science constraints on the trajectory design result from the desire to obtain a particular orbital geometry and are generally not easily quantified. The requirement of the gamma ray spectrometer to obtain low orbits drives the trajectory design through a series of orbits which decrease in radius, and hopefully satisfy all the science requirements on orbit geometry. The general plan is to spend a specified amount of time in a series of circular orbits of predetermined radius before transferring to a lower radius. This satisfies imaging and navigation requirements, keeps the mission on schedule, and requires that only a single general imaging or mapping plan need be developed for any Eros gravity field that may be encountered.

Targeting Strategy

The targeting strategy is simply an algorithm for translating the above spacecraft, mission, and science constraints into a trajectory that can be navigated. The general approach is to develop a broad set of objectives and compute a series of propulsive maneuvers that will steer the spacecraft in a manner that satisfies these objectives. This differs substantially from the

traditional approach of defining a number of constraints and searching for the trajectory that globally minimizes some performance criteria. The NEAR approach is to compute a maneuver that satisfies a local set of constraints and performance criteria and then propagate the trajectory. At the appropriate time, at a minimum of one week in the future, the constraints are reevaluated and another maneuver is targeted. This step-by-step manner continues until all of the science objectives are achieved.

The spacecraft is first placed in an orbit that is in the Sun POS. The solar panel illumination constraint dictates that the spacecraft remain close to this plane throughout the mission. Otherwise, the solar panels would have to be turned too far off the Sun-line when imaging nadir. Furthermore, remaining in this plane for the early part of the mission will minimize the effect of solar pressure on the trajectory and on the attitude control momentum management. This is particularly important at high altitudes where the solar pressure acceleration is a large contributor to the total acceleration. Also, solar pressure is much easier to model when the spacecraft is pointed directly at the Sun.

At the time of arrival at Eros, the spin axis of Eros is pointed toward the Sun. Since the Sun POS coordinate frame z axis points away from the Sun, a retrograde orbit in the Sun POS will be posigrade in the asteroid equator. This is the direction of the initial orbits. As Eros orbits the Sun, the Eros spin axis first points toward the Sun, then perpendicular to the Sun-line, and then away from the Sun. The latitude of the Sun on Eros is shown in Figure 1. The spin axis, which remains essentially fixed in inertial space, appears to rotate in the Sun POS frame. During the early stages of the orbit mission, the spin axis is pointed toward the Sun, a posigrade orbit in the asteroid equator will lie close to the Sun POS and thus be suitable for imaging Eros without turning far off the Sun-line.

As the Eros spin axis aligns perpendicular to the Sun-line, the Sun POS orbit results in a polar orbit with respect to the Eros equator. This is a stable orbit. A problem occurs when the Eros spin axis is around 30° - 60° off the Sun-line. During this stage, the node of the orbit plane with respect to the asteroid equator precesses at a fast rate. It is at this time the spacecraft exits the Sun POS for a "0 phase flyover". This is when the spacecraft flies directly between the Sun and Eros for NIS (NEAR Infrared Spectrometer) observations. A "0 phase flyover" also occurs on approach to Eros, but a second flyover is needed to view the other hemisphere of Eros.

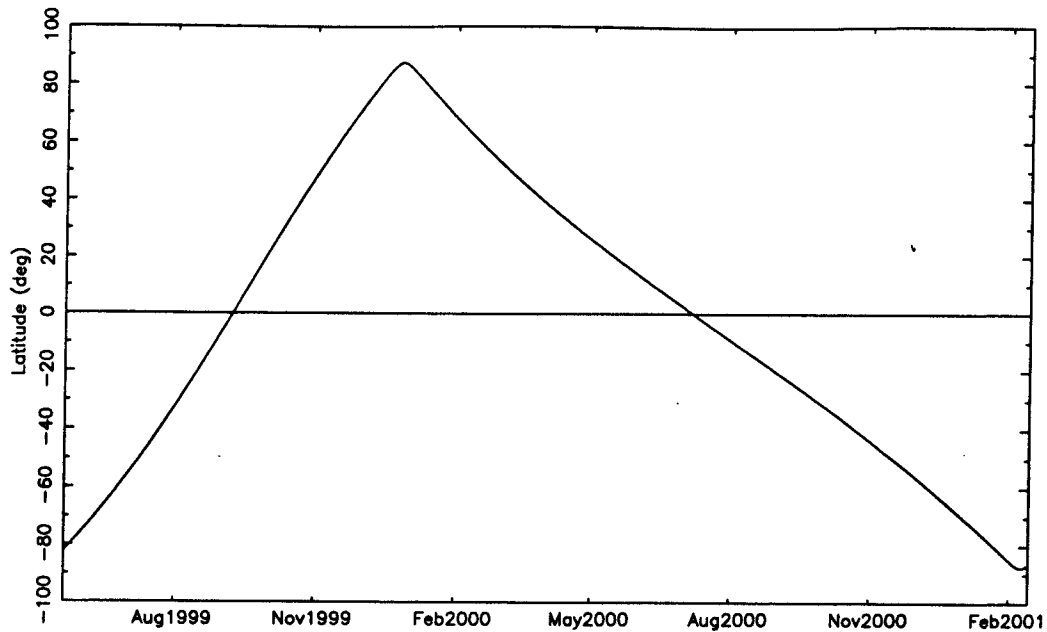


Figure 1. Latitude of the Sun on Eros

Later in the orbit phase, when the spin axis points away from the Sun, retrograde orbits in the asteroid equator will be close to being in the Sun POS. Retrograde equatorial orbits are generally very stable and therefore allow for the orbit altitude to be lowered to 35 km for gamma ray spectrometer observations.

Targeting Algorithm

The trajectory design is accomplished by transforming the targeting strategy into a specific step-by-step procedure referred to as the targeting algorithm. The general approach is to first translate spacecraft and science constraints to geometrical parameters that relate to the spacecraft orbit about Eros. When necessary, propulsive maneuvers are targeted to these orbit-related geometrical parameters and the spacecraft and science constraints are implicitly satisfied. Therefore, the success of the targeting strategy depends on defining geometrical trajectory parameters that relate directly to mission constraints.

The most important geometric trajectory parameters that relate to mission constraints are distances from Eros and angular positions of various celestial bodies with respect to Eros. The classical orbital elements are a convenient and efficient way to describe the size, shape, and orientation of a spacecraft orbit about a central body. As long as the spacecraft acceleration is dominated by the central gravity, the classical orbital elements do not vary significantly, and provide very good targeting parameters. This is true during a large part of the NEAR mission. In high orbits, the solar pressure is a significant perturber relative to the central body gravity, and in low orbits, the gravity harmonics cause the classical orbit elements to osculate. However, we may use the osculating orbit elements as short term predictors of spacecraft motion and thus control the trajectory and satisfy mission constraints by targeting to these parameters.

The radius of periapsis and radius of apoapsis are targeted frequently to control the size and shape of the orbit. The semi-latus rectum and eccentricity may also be used for this purpose. These parameters also implicitly control the period of the orbit. The remaining parameters are angles which describe the orientation of the orbit in inertial space. The longitude of the ascending node, argument of periapsis, and inclination may be targeted in either the Sun POS or asteroid equator orbit frame. Targeting the inclination in the Sun POS coordinate frame is an effective way to satisfy the solar panel illumination constraint. This angle must be kept less than 30° . The asteroid equator coordinate frame is often used to target polar or low inclination orbits. The times that the spacecraft arrives at various points in the orbit, obtained by solving Kepler's equation, are also of interest. The true anomaly, which is the angle measured from periapsis, is included in this category. The times of periapsis, apoapsis, and crossings of the line of nodes or reference planes are all potential maneuver locations. In addition to these classical elements, cartesian components of position and velocity in various coordinate frames may also be target parameters. A combination of cartesian components and classical elements may also be used as potential target parameter sets. A more detailed description of the targeting algorithm is found in Reference 2.

NEAR Trajectory Design

The design of the NEAR orbit phase involves repeated targeting in a step-by-step manner. The model of Eros used for targeting is described in Reference 1. A description of the NEAR mission is described in References 3 through 5. The maneuver schedule resulting from the trajectory design is given in Table 1.

Table 1.
Summary of Revised Eros Orbit Phase ($\mu=4.8 \times 10^{-4} \text{ km}^3/\text{sec}^2$)

Date Time (ET)	DOY	Segment	Orbit (km x km)	Period (Days)	Inc. (deg.) ATE*	Length (Days)	ΔV (m/s)
2/14/00 16:07	45	1	336 x 500	28.4	29	15	10.24
2/29/00 03:07	60	2	499 x 200	21.7	30	10	0.15
3/10/00 07:40	70	3	202 x 198	9.4	37	22	0.40
4/1/00 14:46	92	4	195 x 100	6.0	52	9	0.51
4/10/00 20:03	101	5	101 x 99	3.3	55	12	0.38
4/22/00 15:40	113	6	99 x 50	2.1	60	8	0.47
4/30/00 01:05	121	7	51 x 49	1.2	90	100	2.05
8/8/00 03:56	221	8	51 x 51	1.2	113	19	1.91
8/27/00 23:38	240	9	100 x 50	2.2	109	9	1.14
9/5/00 01:15	249	10	101 x 99	3.3	108	39	0.55
10/14/00 09:35	288	11	500 x 97	17.2	0 phase	10	4.19
10/24/00 10:50	298	12	470 x 200	20.4	130	10	0.99
11/3/00 08:43	308	13	202 x 200	9.5	134	33	0.30
12/6/00 21:05	341	14	205 x 50	4.8	149	7	0.64
12/13/00 22:25	348	15	52 x 50	1.2	151	7	0.97
12/20/00 10:08	355	16	49 x 35	0.9	179	11	1.49
12/31/00 12:30	366	17	35 x 33	0.7	179	31	0.36

Approximate Total Deterministic $\Delta V = 26.74 \text{ m/s}$

*ATE - Asteroid True Equator

The orbit phase begins after a rendezvous burn that slows the spacecraft from approximately 30 m/s to 10 m/s (relative to Eros). This burn occurs on February 2, 2000. A clean-up maneuver is scheduled for February 8, 2000. The spacecraft then flies directly between the Sun and Eros on February 14, 2000. This is the first "0 phase flyover" for NIS observations of the Northern hemisphere. Another "0 phase flyover", during which the Southern hemisphere is observed, occurs midway through the mission. The spacecraft then continues until the approach hyperbola pierces the Sun POS; that is, the plane perpendicular to the Sun-line that passes through the center of Eros. At this point, also on February 14, the Eros Orbit Insertion (EOI) maneuver is executed. This maneuver, performed at a distance of 336 km from Eros, transforms the trajectory from an approach hyperbola to an eccentric ellipse with an apoapsis radius of 500 km. The target parameters are periapsis radius,

apoapsis radius, and an inclination in the Sun POS of 180° . The target parameters are evaluated shortly after the maneuver. The orbit stays very nearly in the Sun POS for one-half of an orbit period, out to 500 km.

When the spacecraft arrives at the 500 km apoapsis radius 15 days later, the first Orbit Correction Maneuver (OCM) is executed to transfer the spacecraft to an orbit with a periapsis radius of 200 km. At 200 km radius, OCM 2 is performed and the orbit is circularized.

After 22 days, or approximately two and a half orbits at the 200 km radius, an OCM is executed to lower the spacecraft into a transfer orbit with the periapsis at 100 km. This maneuver is placed at a nodal crossing in the Sun POS. As a result, the Sun POS inclination is targeted to 180° . Recall that the Sun POS coordinate frame rotates as Eros orbits about the Sun and the orbit plane tends to remain fixed in inertial space. In order to keep the spacecraft in the Sun POS, the maneuver must be placed in the Sun POS. In order to maintain at least seven days between maneuvers, the spacecraft remains in the transfer orbit for 9 days, or one and a half orbits. At the periapsis of 100 km, another OCM is performed and the spacecraft orbit is circularized.

The projection of the spacecraft orbit into the Sun POS coordinate frame is shown in Figure 2 from EOI through the 100 km orbits. The view is from the Sun looking toward Eros. The spacecraft moves in a counter clockwise (prograde) direction while the Eros rotation is also counter clockwise.

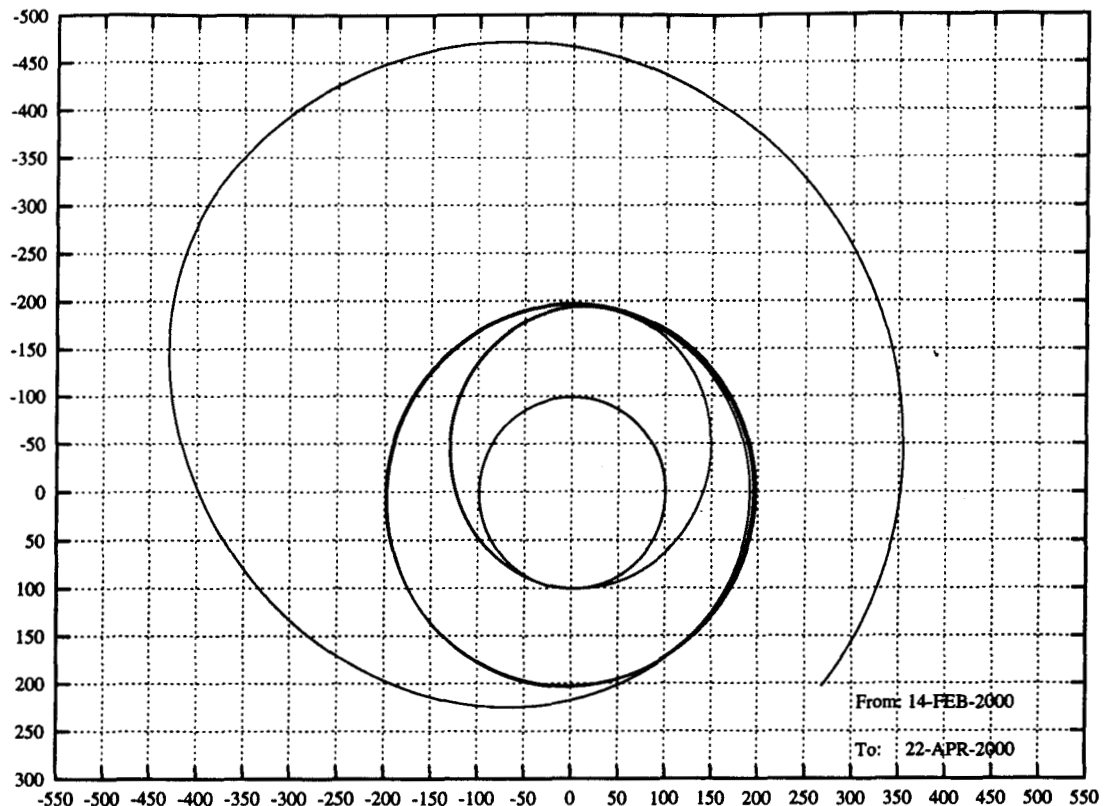


Figure 2. EOI through 100 km Orbits in Sun-Plane-of-Sky

After 12 days in a circular 100 km orbit, an OCM is performed to place the spacecraft in a transfer orbit with the periapsis at 50 km. It remains in the transfer orbit for 8 days before circularizing at 50 km. This OCM on April 30 targets to a polar orbit in the Eros equator. The polar, circular, 50 km orbit is very stable, allowing the spacecraft to remain in this orbit for 100 days with very little precession out of the Sun POS. The spacecraft could, in fact, remain in this orbit for even longer. A power margin constraint, however, does not allow this.

Because Eros is at aphelion during this time, the solar panels cannot be turned more than 20° off the Sun-line to maintain a 5% power margin. This is even more restrictive than the 30° constraint in effect for the rest of the mission. When the spacecraft orbit becomes very close to 20° off the Sun-line, an OCM is performed to place the orbit back into the Sun POS. This OCM on August 8 is performed at a nodal crossing in the Sun POS coordinate frame. The OCM switches the direction of the node from descending to ascending. Consequently, the orbit inclination immediately following the OCM is still 20° off the Sun-line, but now the precession moves the orbit plane in the direction toward the Sun POS rather than away from it.

This OCM affects the inclination in the Eros equator as well. The spacecraft remains in a 50 km circular orbit, but it is no longer polar. Recall

that the Sun POS coordinate frame rotates as Eros orbits the Sun. As a result, the spacecraft is now in a 113° inclination with respect to the Eros equator. This is the first time in the mission the spacecraft flies in a retrograde orbit. The spacecraft remains in this orbit for 19 days until the next OCM is performed on August 27. This OCM puts the spacecraft in a transfer orbit with an apoapsis of 100 km.

Figure 3 shows the projection of the spacecraft orbit in the Sun POS coordinate frame from the 100×50 km transfer, through the 50×50 km orbits, and back to the 100×50 km transfer orbits.

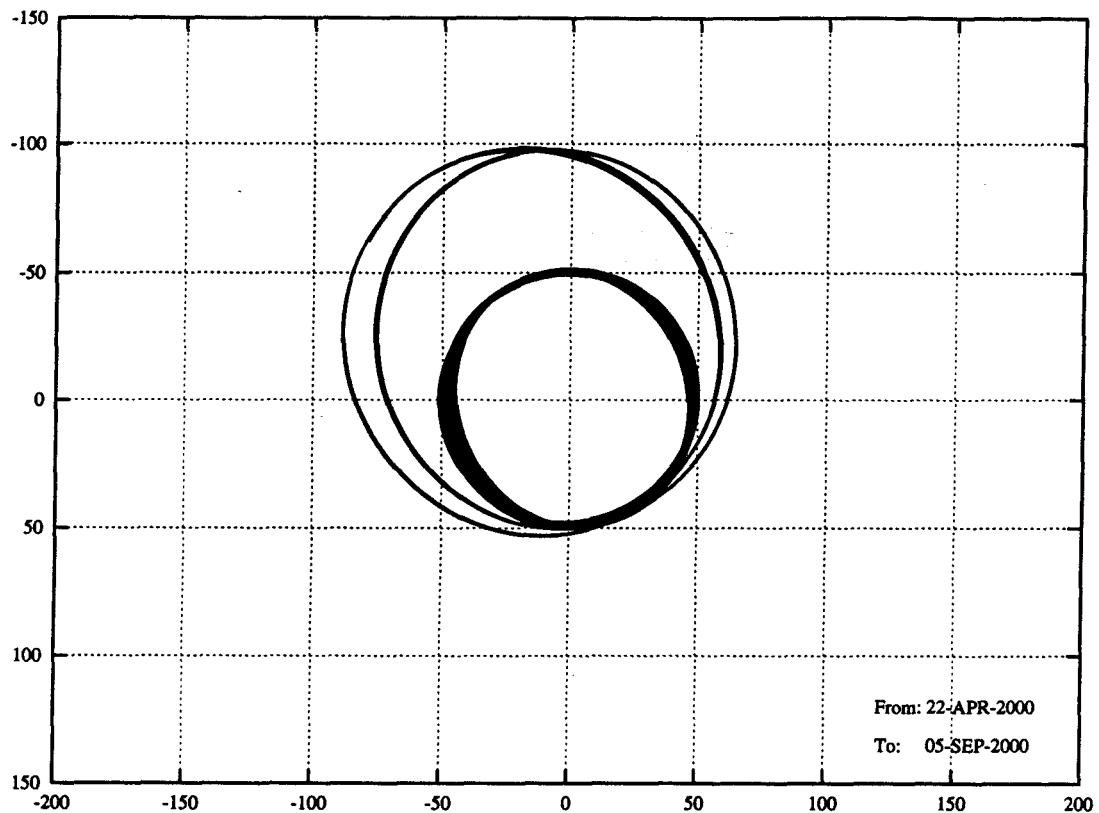


Figure 3. 100×50 km Transfer Orbits, 50×50 km Orbits, 100×50 km Transfer Orbits in Sun-Plane-of-Sky

The orbit is circularized at 100 km on September 5. The spacecraft remains in a 100 km circular orbit for 39 days.

A primary science objective is to obtain infrared images of Eros at low phase angle. This is accomplished once during the approach and rendezvous with Eros in February 2000. The approach asymptote is targeted to a point directly between the Sun and Eros and the spacecraft imaged the northern hemisphere of the asteroid. In October of 2000, the southern hemisphere is pointing toward the Sun. In order to obtain images at a very low phase angle, it is necessary to place the spacecraft in an orbit that passes over the subsolar

point. This is accomplished by targeting the inclination in the Sun POS coordinate frame to 90° . The apoapsis radius is targeted to 500 km, which gives the orbit a period of greater than 2 weeks. The actual 0 phase flyover occurs at a distance of approximately 190 km.

After 10 days, the spacecraft crosses the Sun POS at a distance of approximately 470 km. At this point, an OCM is targeted to an elliptical orbit with a periapsis distance of 200 km in the Sun POS. This is a critical maneuver. If for some reason this OCM is not performed, the spacecraft will continue on the overfly orbit which would fly behind the asteroid. This would violate the very important spacecraft constraint to keep the solar panels illuminated.

The spacecraft remains in a 470×200 km orbit for 10 days or half of a revolution around Eros. At the periapsis point of 200 km, an OCM is executed to circularize the orbit. The spacecraft stays in a 200×200 km orbit for 33 days.

Figure 4 shows the projection of the spacecraft orbit in the Sun POS coordinate frame from the circular 100 km orbit, through the 0 phase flyover back into the Sun POS, the 470×200 transfer orbit, and the 200 km circular orbits. Figure 5 shows the same set of orbits from the north ecliptic view.

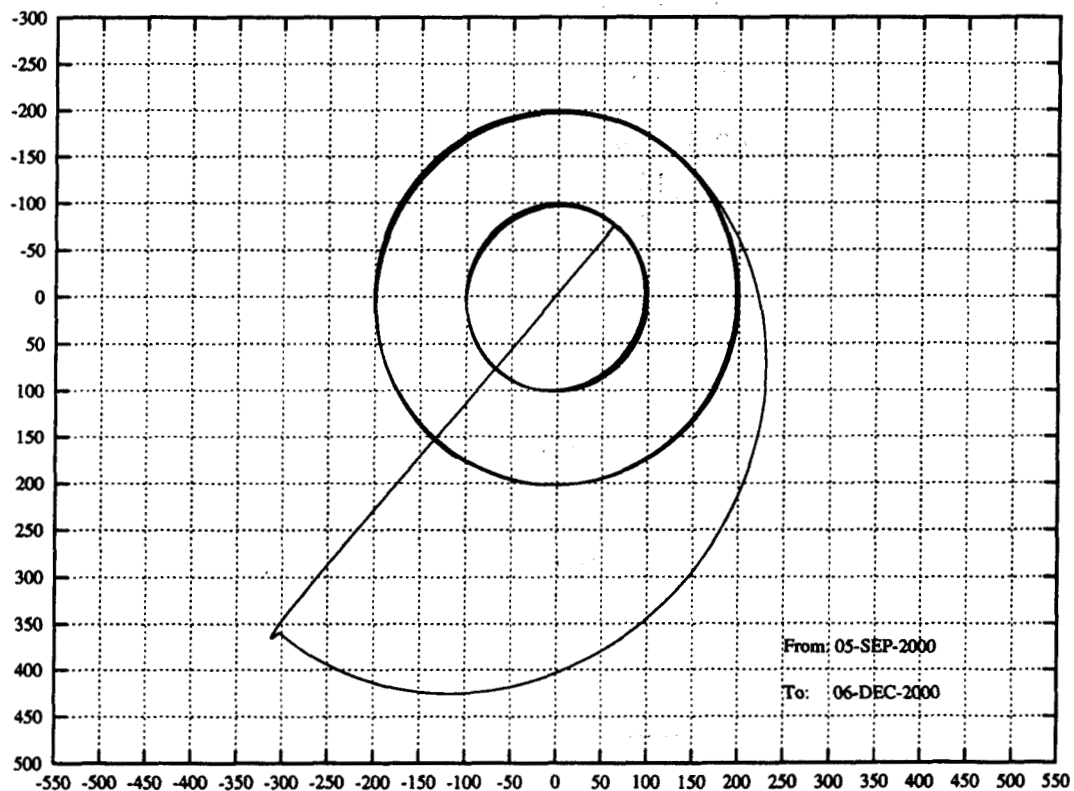


Figure 4. 100 x 100 km Orbits, 0 Phase Flyover, Transfer Orbit, 200 x 200 km Orbits in Sun-Plane-of-Sky

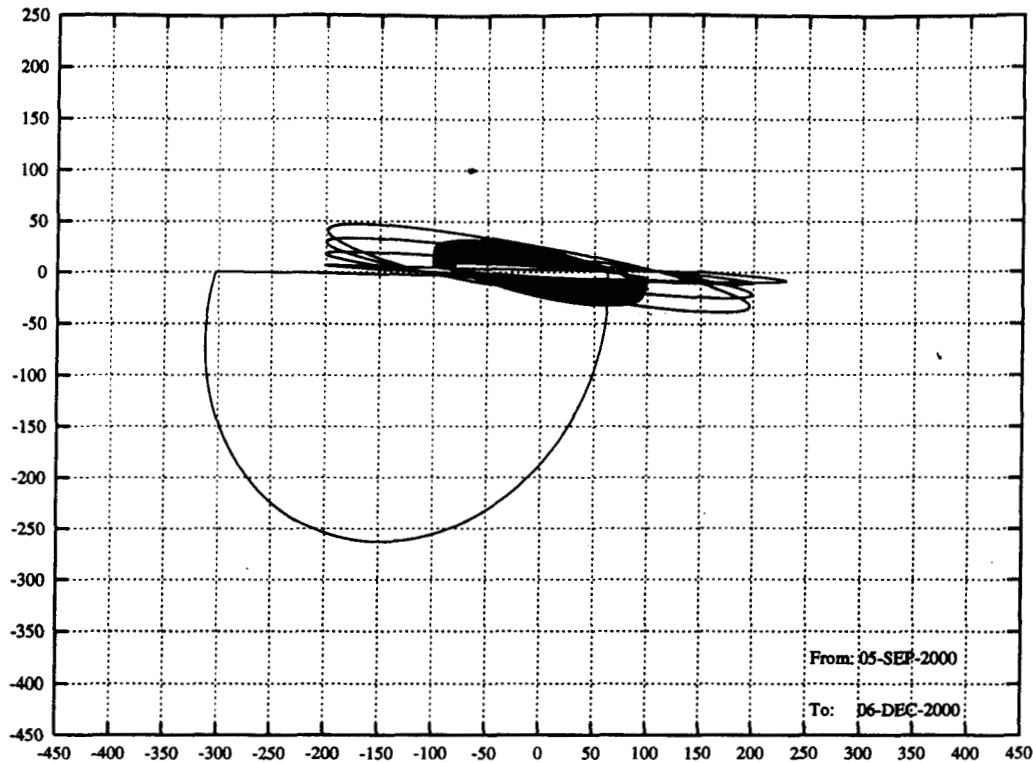


Figure 5. 100 x 100 km Orbits, 0 Phase Flyover, Transfer Orbit, 200 x 200 km Orbits in Polar Ecliptic View

Another very important science objective is to obtain low altitude gamma ray and x-ray spectrometer measurements of Eros. From a 35 km orbit, Eros fills the field of view and long integration times are required to obtain the data needed to characterize the composition of Eros. Consequently, the remainder of the mission is dedicated to achieving low orbits to satisfy this science objective.

On December 6, an OCM is executed to place the spacecraft in a transfer trajectory with a periapsis radius of 50 km. The spacecraft remains in this orbit for 2 1/2 revolutions around Eros, or about 7 days. At the periapsis radius of 50 km, another OCM is performed to circularize the orbit. The spacecraft is in this orbit for another 7 days before it begins in another elliptical transfer orbit (50 x 35 km). The spacecraft then circularizes at 35 km on December 31. This orbit is a retrograde equatorial orbit about Eros. The spacecraft remains in this orbit for at least a month to obtain x-ray and gamma ray observations. Following this, low altitude and landing maneuvers are performed to complete the mission (Reference 6).

Figure 6 shows the projection of the spacecraft orbit in the Sun POS coordinate frame from the 200 x 50 km transfer orbit through the 35 km circular orbits.

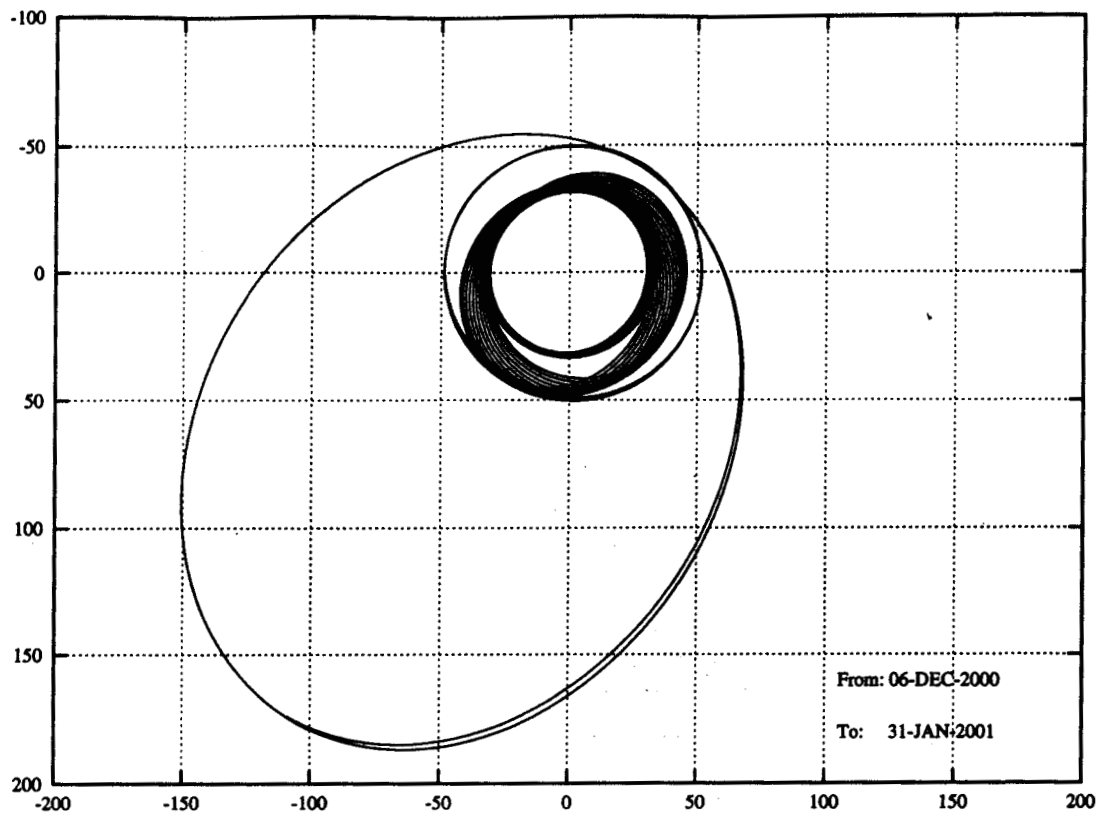


Figure 6. 200 x 50 km Transfer Orbits, 50 x 50 Orbits, 50 x 35 km Transfer Orbits, 35 x 35 km Orbits in Sun-Plane-of-Sky

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